[redacted]

CLL 10108

Copy No. ________

CLASSIFICATION CHANGE

To UNCLASSIFIED
By authority of GDS-EO 11652 GP-4
Changed by O. L. Merritt Date 12/3/72
Classified Document Master Control Station, NASA
Scientific and Technical Information Facility

Grumman FINAL REPORT

FEASIBILITY STUDY OF UTILIZATION OF LM FOR PROJECT ABLE [redacted]

submitted to

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
MANNED SPACECRAFT CENTER
HOUSTON, TEXAS 77058

and

GEORGE C. MARSHALL SPACE FLIGHT CENTER
HUNTSVILLE, ALABAMA 35812

by

GRUMMAN AIRCRAFT ENGINEERING CORPORATION
BETHPAGE, NEW YORK 11714

[redacted] contains information affecting the [redacted] United States [redacted] meaning of the [redacted] Sections 793 and [redacted] revelation [redacted] in any manner [redacted] is prohibited by law.

[redacted]

December 12, 1966
Contract NAS 9-6209

Volume I
Technical Summary

[redacted]

[redacted]

PREFACE

This document completes the reporting requirements of NASA/MSC Contract NAS 9-6209, "An Effort Pertaining to Project ABLE". The study was performed under the technical direction of NASA/MSFC. The objective of the 110 day effort was to establish overall systems approaches, including technical and programatic feasibility of solar reflectors* operating in Earth orbit, supported by the LM vehicle. The work effort consisted of a 90 day technical study and program planning period and a 20 day final report preparation period. Progress reviews were held, at the request of the COR, on September 7, October 6, October 11, November 6, and November 13, 1966. In order to accommodate this increased number of reviews, a two week extension in delivery of this final report was granted by the COR.

The final report consists of three volumes, bound into four documents. The first document, "Volume I - Technical Summary", briefly outlines the objectives of the study, summarizes the results, and gives conclusions and recommendations for further study. Areas of advanced technology, wherein further effort is recommended based on the study results, are also described. The second document, "Volume II - Technical Report" is a comprehensive condensation of the work performed during the study period. The third document, "Volume III, Part 1 - Program Plan", consists of five sub-plans, covering Development and Acceptance Test, Pre-Launch Operations, Product Support, Manufacturing, and Reliability. The fourth document, "Volume III, Part 2", contains the sixth sub-plan, Cost.

*The term "reflector" is properly applied to the thin membrane of metallized plastic which actually reflects the sunlight. In this report, however, the term reflector should be interpreted as either the membrane itself or including the support structure.

[redacted]

[redacted]

TABLE OF CONTENTS

1.0 INTRODUCTION

2.0 CONCLUSIONS & RECOMMENDATIONS

2.1 MISSION

2.2 CONFIGURATION

2.3 RECOMMENDATIONS FOR FUTURE STUDY

3.0 MISSION ANALYSIS

3.1 ILLUMINATION

3.2 ORBITAL MECHANICS

3.3 FLIGHT OPERATIONS

4.0 REFLECTOR/SUPPORT STRUCTURE

4.1 MATERIALS

4.2 STRUCTURAL ANALYSIS

4.3 THERMAL ANALYSIS

5.0 ABLE LM

5.1 THERMAL ANALYSIS

5.2 REACTION CONTROL

5.3 GUIDANCE & CONTROL

5.4 ELECTRICAL POWER

5.5 ENVIRONMENTAL CONTROL

5.6 COMMUNICATIONS

5.7 INSTRUMENTATION

5.8 CONTROLS & DISPLAYS [redacted]

5.9 CREW PROVISIONS

[redacted]

1.0 INTRODUCTION

The use of a large reflector in Earth orbit to illuminate selected geographical areas during the hours of darkness has been proposed. The objectives of the present study were to determine the technical feasibility of this concept, and to furnish system trade-off and design data integrating the LM spacecraft and/or its subsystems and components with the reflector. An additional objective of this study was to select a conceptual design and provide a program plan that would permit determination of the schedule and cost feasibility of the concept.

A list of the major groundrules which were supplied to Grumman by MSFC follows:

- Design emphasis shall be placed on the maximum effective utilization of existing, modified, or planned LM hardware.
- Overall technology constraints shall be based on attainment of hardware which would be operational in the 1968-69 time frame.
- The minimum illumination level which would be considered acceptable would be equivalent to a 400 ft. diameter ideal reflector at synchronous altitude.
- Illumination shall be required during all periods of darkness at the target.
- Reflector sizes ranging from 400 ft. to 3000 ft. shall be considered.
- Orbits with nominal altitudes of 6000 nm. and 24 hr. synchronous shall be of primary interest.
- The degree to which man can enhance the program by his in-space participation shall be determined.
- Single and dual Saturn V launches shall be considered.

1.0 INTRODUCTION (continued)

- The minimum operating lifetime shall be at least six months without resupply. However, an operating lifetime of 12 months is highly desirable with possible resupply.
- Useful payload weights and ambient environment data will be provided by NASA.

Using these groundrules and the assumptions listed in the various sections of the report, the technical feasibility of the Project ABLE concept was studied.

2.0 CONCLUSIONS AND RECOMMENDATIONS

2.1 MISSION

2.1.1 Altitude - Deflection Considerations

Target area illumination is extremely sensitive to deflection of the reflector surface. For example, if a 1000 ft diameter reflector could be designed which remained flat to ± 1 ft (very tight tolerance), illumination equivalent to a full moon could be achieved only by selecting an orbital altitude of 3-5000 n.mi. Higher altitudes would require tighter tolerances, large numbers of reflectors, or acceptance of less illumination.

The tight tolerances suggest that low altitudes are desirable, resulting in the requirement for more than one reflector, if illumination is to be provided continuously to the target. At 3-5000 n.mi., approximately 4 reflectors are needed for continuous illumination.

2.1.2 One/Two Sides Reflective

Orbital mechanics studies indicate that changes in rotational rate can be minimized if both sides of the reflector surface have reflecting capability. The high temperatures which result, however, would eliminate Mylar from consideration. Kapton could be used, but it is a relatively new material for this space application. It is suggested that additional studies be done to advance the state-of-the-art of Kapton technology. The use of Kapton would permit reflectorizing both sides of the membrane, thus eliminating the constraints on trajectory and target choices that result from a reflector with one side coated.

2.1.3 Manned Vs Unmanned Mission

The evaluation of the need for man to aid in implementing the ABLE mission concept has resulted in the following preliminaries conclusions:

- Man should be considered for the first ABLE flight(s)
- Man's value on subsequent flights requires further study.

This is mainly due to the fact that some of the more intricate operational phases of the ABLE mission, such as reflector deployment, will be difficult to duplicate during the development ground test program.

2.1.3 cont'd

A program concept permitting a delayed decision on manned vs unmanned ABLE flights is therefore recommended. This would mean that the ABLE spacecraft should be designed so that it could be flown as part of either a manned or unmanned flight.

2.1.4 Radiation

The major problem in evaluating the potential radiation hazard for the ABLE mission is the uncertainty in the predicted 1968-69 environmental model. Unfortunately, it is unlikely that any new emperical data will be available early enough to provide statistical verification of any assumed model environment before major program decisions have to be made to meet a 1969 launch date. Despite the uncertainty in the radiation environment, the following preliminary conclusions have been reached:

- The use of mylar for the reflector is marginal at altitudes below synchronous
- There are no major problems while the crew is in the spacecraft
- At best, limited extra-vehicular activities may be possible.

2.1.5 Weight

The maximum ABLE payload which can be carried on a manned Saturn V launch is restricted by the maximum weight that can be supported in the SLA which is about 32,000 lbs. The estimated weight of an ABLE LM exclusive of the reflector and its support structure is 9700 lbs. This means that, for a spoke and torus reflector configuration, the maximum reflector size that can be carried with a manned ABLE LM is about 1200 ft. in diameter.

For an unmanned Saturn V launch, the maximum allowable payload is governed by the launch vehicle stack limit (above the IU). For this case, reflector diameters of 3,000 ft or greater could be carried in addition to an ABLE LM.

[redacted]

2.2 CONFIGURATION

2.2.1 Reflector/Support Structure

Three families of reflector/support structure were studied, with each family defined by the location of the support structure relative to the reflector surface, and by the method of deployment. After consideration of such factors as weight, ease of deployment, sensitivity to deflections, packaging efficiency, ease of resupply, and testing problems, it was felt that a two dimensional shape such as a torus with four spokes appeared most feasible. Refer to Figure 2.2-1.

Methods of deployment were grouped into mechanical and inflation techniques, and here the physical simplicity of inflation and its adaptability to various sizes of the support structure, made it a favored method.

There did not appear to be any greater problem in integrating the LM spacecraft with any type of reflector configuration because of the nature of the interfaces between the LM and the reflector/support structure.

2.2.2 ABLE LM

The use of the basic LM vehicle and related subsystems/components, in conjunction with a large reflector, permits utilization of a developed system which offers the following advantages:

- o Operational in the 1968-1969 time period
- o Docking capability with the Apollo CSM
- o Compatible with Saturn V launch environment
- o Mounting in the spacecraft LM adapter (SLA)
- o Available subsystems compatible with both manned and unmanned operation.

The ABLE version of LM is almost entirely composed of developed LM hardware. The only major new items are radiators, solar arrays and rechargeable batteries, and low-level thrusters. Major items removed from LM include the propulsion subsystems, the landing gear, and the abort guidance and radars. These changes are shown in Table 2.2-1 and Figure 2.2-2.

An examination of the new items indicates that all but the thrusters result from six-month mission duration requirements. It is of interest to note that if an engineering test flight of 3 - 14 days were considered, in order to checkout deployment and flatness, an absolute minimum of new equipment would be required.

[redacted]

[redacted]

REFLECTOR CONFIGURATIONS

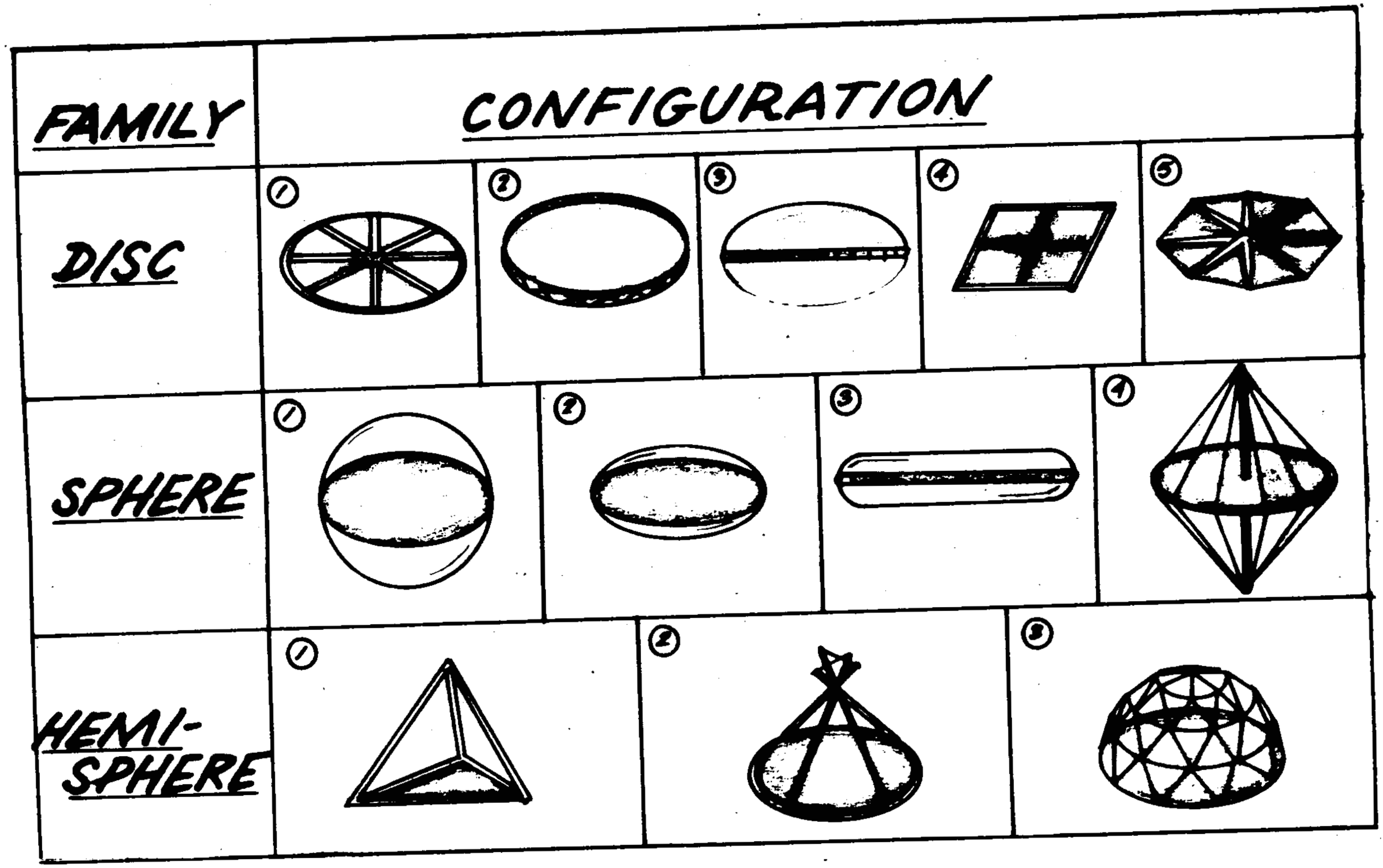


FIGURE 2.2-1

[redacted]

[redacted]

Table 2.2-1

ABLE LM WEIGHT STATEMENT

LM - AS-504 CONTROL WEIGHT - TLI	32,000
Ascent Structure	
Unchanged	
Descent Structure	
Remove:	
Base Heat Shield	-146
Add:	
ABLE Canister & Supports	1,000
LM/Reflector Attachment Penalty	100
Guidance, Navigation & Control	
Remove:	
DECA	-7
GDA	-6
ASA (AGS)	-32
AEA	-20
RGA	-2
DEDA	-8
AOT	-22
Rendezvous Radar	-75
Landing Radar	-38
Added:	
PCA	60
LOTS	24
Crew Provisions	
Unchanged	
Environmental Control	
Add:	
Fixed radiators (Installed Wt.)	50
Landing Gear	
Remove:	
Total Landing Gear	-465

[redacted]

Instrumentation

Same weight as Apollo/LM mission

Electrical Power

Added:

Deployable Solar Arrays Rechargeable Batteries (2) ECA's (2) Battery Chargers & Voltage Regulators Wiring, Supports, etc.	1200

Propulsion

Ascent Propellant - Total	-5090
Descent Propellant - Total	-17,749
D/S Propellant Tanks	-440
A/S Propellant Tanks	-172
D/S Engine	-377
A/S Engine	-212
A/S Helium	-13
A/S Helium Tanks	-111

Added:

D/S Supercritical Helium Tank	100
Helium	55
Plumbing & Supports	15

Reaction Control

Unchanged from LM

Communications

Remove:

TV	-8
S-Band Erectable & Steerable Antenna	-38

Add:

VHF Antenna	3
DCA & CMD Rcvr	40

Controls & Displays	70
TOTAL LM	9686
1100 ft. diameter reflector	17,400
TOTAL ABLE LM	27,086
GROWTH ALLOWANCE	4914
	32,000 lbs.

[redacted]

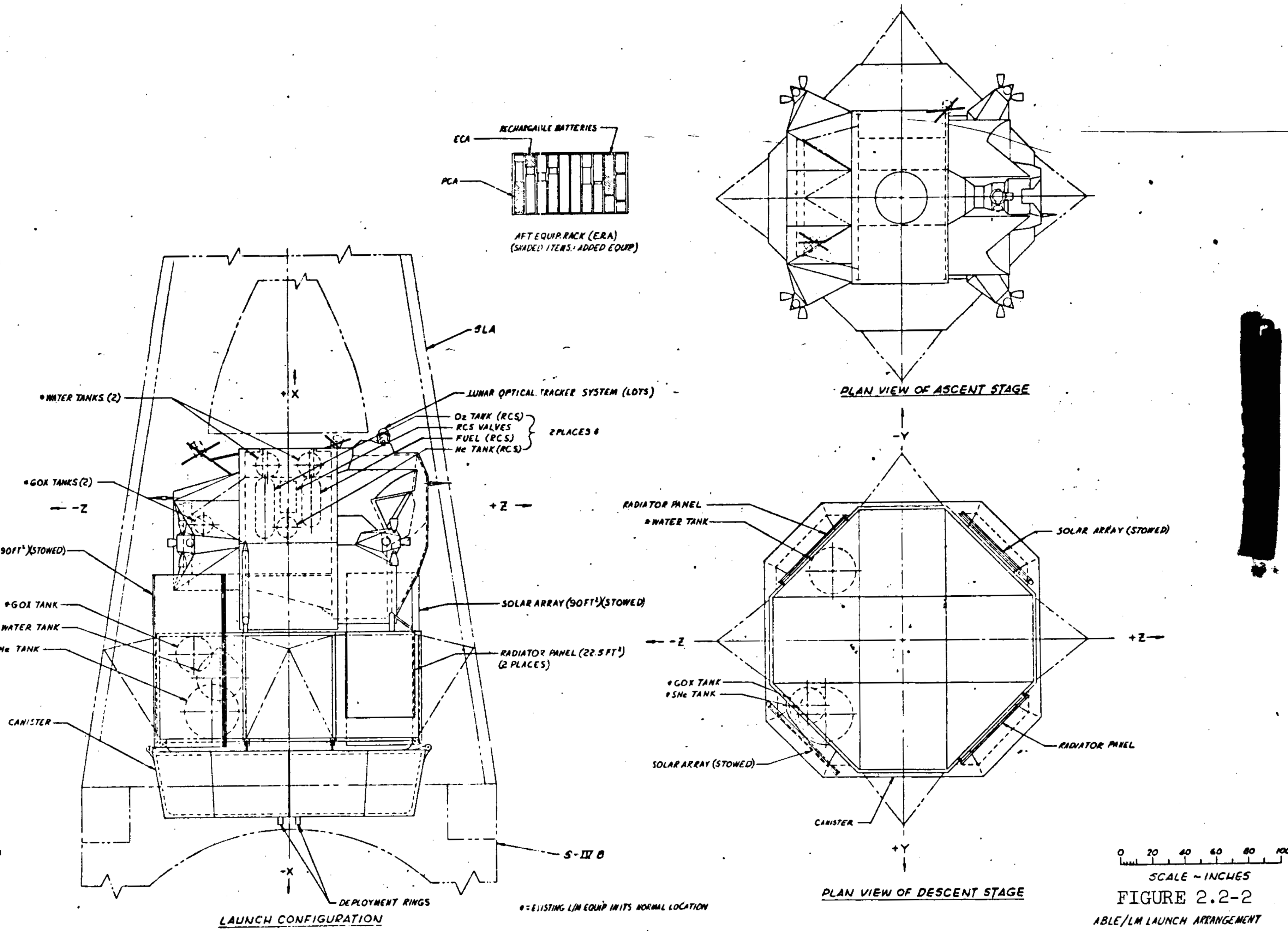


FIGURE 2.2-2
ABLE/LM LAUNCH ARRANGEMENT

[redacted]

2.3 RECOMMENDATIONS FOR FUTURE STUDY

2.3.1 Mission Requirements

During the study, questions continuously arose as to the minimum acceptable mission requirements for use in spacecraft design. In particular, the required intensity of the illumination, and the duration of time during the night that it must be provided, had major impact on the study. Consequently, better definition of the mission requirements, both with respect to the stated ABLE military objective and to other possible applications, would permit more definitive studies to be performed in the future.

2.3.2 Deflections

Because of the major effect that reflector flatness has on illumination, and as a result of the difficulty in ground testing a full scale configuration, it is recommended that techniques for adjustment of the reflector surface in orbit, by either automatic or manual means, be given special emphasis in future studies.

2.3.3 Structural Data

Little test data exists on the behavior under load of large inflated structures. Extrapolations from existing data were used wherever possible. Parameters such as damping factors were extremely difficult to estimate. It is therefore recommended that additional data be generated, by tests of either a scale model, or of a segment of a full size configuration.

2.3.4 Kapton Technology

The characteristics of Kapton are similar to those of Mylar, except that Kapton can tolerate significantly greater thermal extremes and radiation doses than Mylar. The newness of Kapton, in an application such as ABLE, makes the continued study of Kapton characteristics and fabrication techniques worthwhile.

[redacted]

3.0 MISSION ANALYSIS

3.1 ILLUMINATION

To provide the required illumination over a particular area of interest, consideration must be given to several important factors. These factors include altitude of operation, reflector reflectivity and area, atmospheric attenuation, reflector system geometry and, most important, reflector deflection from an optically flat surface. A detailed discussion of these factors is provided in Appendix A, "Optics Analysis." In this section, only the more important conclusions, which can be drawn from Appendix A, will be discussed.

The sunlight reflected from a flat reflector in orbit is contained in a cone with approximately a 0.5 deg cone angle, which is the apparent angle of the sun seen from the earth. The illuminated area on the ground is therefore dependent only on the altitude of the spacecraft. If the reflecting surface departs from a plane, however, the reflected cone of light broadens, thus covering more ground area but with reduced average illumination. This surface deflection was quantified by defining it as the center distance between a spherical cap and the plane defining its edges.

In order to appreciate the interrelationship between these four variables (attitude, reflector diameter or area, deflection, and resulting ground illumination), Figure 3.1-1 was generated. Two values of ground illumination were selected, corresponding to "full moon" and "min. illum." (0.1 full moon). Preliminary analysis indicated that deflections of the order of 1-10 feet might be achievable, and so this range was plotted. Finally, the values of illumination were reduced by 50% to account for realistic values of reflectivity, atmospheric attenuation, and angle of incidence of the sunlight.

The great dependence of illumination on altitude, reflector diameter, and deflection is graphically shown in Figure 3.1-1. For example, if it were possible to construct a 1000 foot reflector, which deflected more than 1 foot at the center, achievement of 0.1 full moon would require orbiting at approximately 10,000 nm or lower. Achievement of full moon illumination

[redacted]

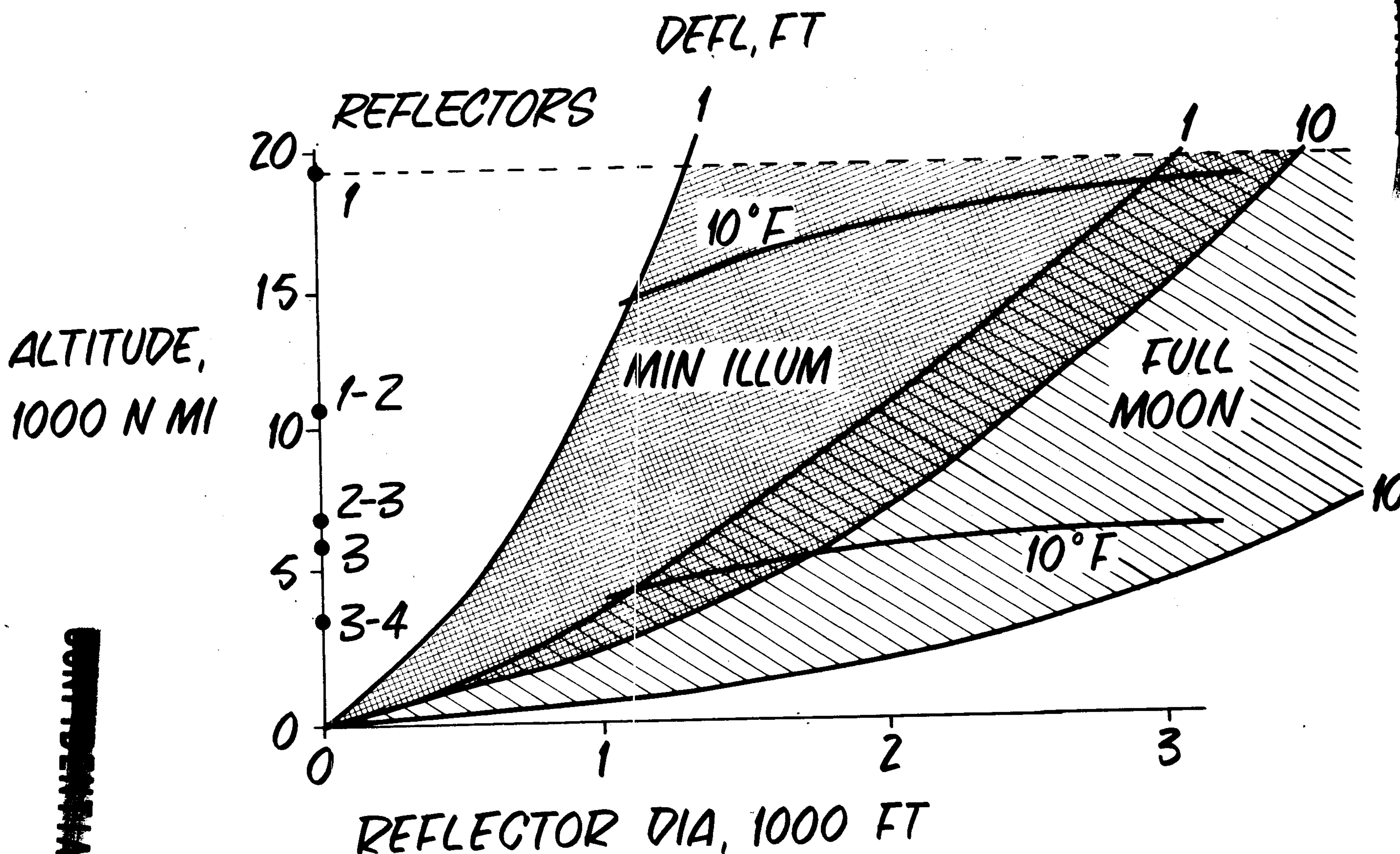
would similarly require orbiting at 5,000 nm or lower. One major implication of these orbits is the necessity for multiple reflectors, time-phased in the orbit, in order to provide illumination during the entire night. The problems of packaging, mounting launching and deploying 3-4 or more reflectors requires further study.

Examination of Figure 3.1-1 also reveals the difficulty of increasing illumination by increasing reflector diameter. The study indicates that the major causes of reflector deflection are due to thermal distortions and manufacturing tolerances. The nature of the thermal distortion is that it increases with the square of the reflector diameter as does the area. The rate of increase depends, among other things, on the temperature gradient in the support structure. A 10° gradient, which is about as good as can be achieved, is plotted for both full moon and minimum illumination. The horizontal nature of both curves indicates that for fixed illumination, the increases in diameter do not cause increases in allowable altitude. In other words, the advantages of increases in area which result from increasing diameter are effectively eliminated by larger deflections. Since manufacturing errors will also increase as some power of the diameter, a similar conclusion is likely.

[redacted]

FIGURE 3.1-1

ALLOWABLE DEFLECTION



[redacted]

3.2 ORBITAL MECHANICS

3.2.1 Circular Orbits (Unperturbed)

In order to be able to predict control system requirements for the reflector, attitude timelines have been generated, showing the Right Ascension and Declination of the unit normal of the reflector relative to an earth-centered inertial coordinate system. Based on analysis, the "worst case" conditions were found to occur at the winter solstice.

When the target is not covered, the reflector is assumed to undergo a constant slew rate so as to be in position for the next period of coverage. Slew rates between coverage regions were chosen on the basis of minimizing the rotational rate change requirements (and hence, propellant requirements) on the control system. Reflecting surfaces on both sides of the plate are desirable for minimum control propellant expenditures. Based on these preliminary studies, it is concluded that, in order to enhance flexibility in mission planning, the reflector should be coated on both sides if the material technology permits.

3.2.2 Stable Orbits

Perturbations of circular orbits were studied, and it was seen that the two most important perturbations results, eccentricity and apside location, exhibited cyclic behavior. After 180 days, the eccentricity reached a maximum value and after one year returned to zero. Similarly, the line of apsides advanced for the first half year and regressed back to the initial value during the second half. There appears to be some value of eccentricity which yields an orbit which remains unchanged relative to the target at midnight under the influence of the perturbing forces considered. With such stable orbits, stationkeeping requirements can be greatly reduced, yielding significant savings in propulsion system redesign and total system weight.

The capability of the LM RCS system to perform stationkeeping maneuvers was evaluated. After deployment of the reflector, it is estimated that the LM RCS would have 500 lb. of propellant available for stationkeeping maneuvers. This would then provide a Δ V capability of 162 ft/sec, with which the parameters inclination, period and eccentricity can be corrected.

3.2.3 Conclusions and Recommendations

Based on the above parametric studies, the following conclusions pertinent to the Able mission are cited:

- The minimum altitude at which one satellite provides full nightly coverage is approximately 11,000 n.mi. (12 hr. orbit).
- Inclined orbits reduce the percentage time of the year in which loss of coverage due to occultation by the earth's umbra exists.
- The length of time per orbit in which the reflector is occulted increases with increased orbit altitude.
- Stationkeeping requirements associated with maintaining a favorable circular orbit decreases with increased altitude.
- Stable orbits exist which theoretically eliminate the requirements for stationkeeping.
- Circular orbits will yield slightly higher illumination than stable orbits of equal period.

Although a 12 hr. orbit yields coverage from night to morning twilight, illumination levels during the early and late periods of the night will be severely degraded. Hence, the likelihood of a multiple reflector system for orbits with resonent periods other than synchronous is high. In addition, the certain occultation of the reflector at all altitudes supports the requirement for multiple satellites. For a 12 hr. orbit inclined 28.5 degrees, occultation periods of up to .94 hrs/orbit for up to 52 days is possible.

The stationkeeping requirement for maintaining an effective circular orbit was found to be large. Additional LM RCS tankage or use of the LM ascent propulsion tanks would be necessary. Hence, it is recommended that the stable orbit concept be adopted for the Able mission. Further study to determine the effects of off-nominal characteristics of the reflector (i.e. degradation in reflectivity) on maintaining the orbit should be performed to accurately determine the extent of stationkeeping required.

[redacted]

3.3 FLIGHT OPERATIONS

3.3.1 Ground Communication Coverage

For initial planning purposes, the Apollo Manned Space Flight Network (MSFN) has been investigated for fulfilling the primary ground communications and tracking requirements for Project ABLE. In addition, the possibility of using the Satellite Control Facility (SCF) as a secondary network is considered. The SCF support capability is based on the defense objective of the ABLE mission and the proposed incorporation of the Space Ground Link Sub-system (SGLS) into the SCF sites.

As a result of the orbiting reflector's high altitude, full MSFN ground communications are possible; thus providing full time command capability if reequired. In the event that rendezvous and resupply with a second vehicle is required, ground coverage, in conjunction with the MSFN capabilities, permit adequate support of both vehicles.

As indicated from study of the SCF network, continuous visibility is not available; however, the no-coverage area presented to the 28.5° inclination orbit is considered to be tolerable. For the ABLE elliptical reference orbit, continuous coverage is possible in the vicinity of Southeast Asia.

The "Dual" SCF support in the Southeast Asia area, accomplished by two sites affording simultaneous coverage (Guam and Mahe Seychelles), is marginal at 6,000 n.mi. altitude. For the elliptical reference orbit, the visibility becomes greatly expanded so that multiple SCF site support is assumed.

In comparing the SCF composite visibility to that of the MSFN, the latter appears to have only a slight advantage. The significant feature of the MSFN network, however, is the general increase in operational capability produced by the greater number of stations.

[redacted]

3.3.2 Operation Launch Vehicle Consideration

~~CONFIDENTIAL~~

TABLE 3.3-1

Summary of Alternative Saturn Launch Vehicle Advantages and Disadvantages

	Single Saturn V Direct Ascent	Single Saturn V Parking Orbit	Dual Launch Saturn V	Dual Launch Saturn V & IB Parking Orbit Rendezvous
Economical	x	x		
Large Orbital Payload		x	x	x
Simplified Launch Pad Activities	x	x		x
Orbital Operations Similar to Apollo		x	x	
S-IVB Present Restart Capability Sufficient	x			
Orbital Operations Simplified	x	x		
CSM Not Required to Inject	x	x	x	x

Notes:

x -- Advantage

~~CONFIDENTIAL~~

[redacted]

3.3.3 LM/Reflector Test Flight

Consideration has been given to conducting an early Saturn IB launched LM/Reflector test flight to demonstrate basic Reflector deployment and flight operational capability. The purpose of this mission, an orbital test of the LM/Reflector, could be achieved with minor modifications to a basic LM and with little impact to the present NASA launch schedule. This mission would be similar to the currently planned Apollo mission "258", a dual Saturn IB launch of an unmanned LM and manned CSM.

The short duration, low-altitude flight to test Reflector deployment and operation could be flown using a LM Reflector and S-IB launch vehicle for an unmanned LM/Reflector launch into earth orbit. This launch would be planned in conjunction with an Apollo Applications mission so that the CSM and crew could then rendezvous with the LM/Reflector for the deployment check. The manned CSM would be utilized only after its main mission was completed.

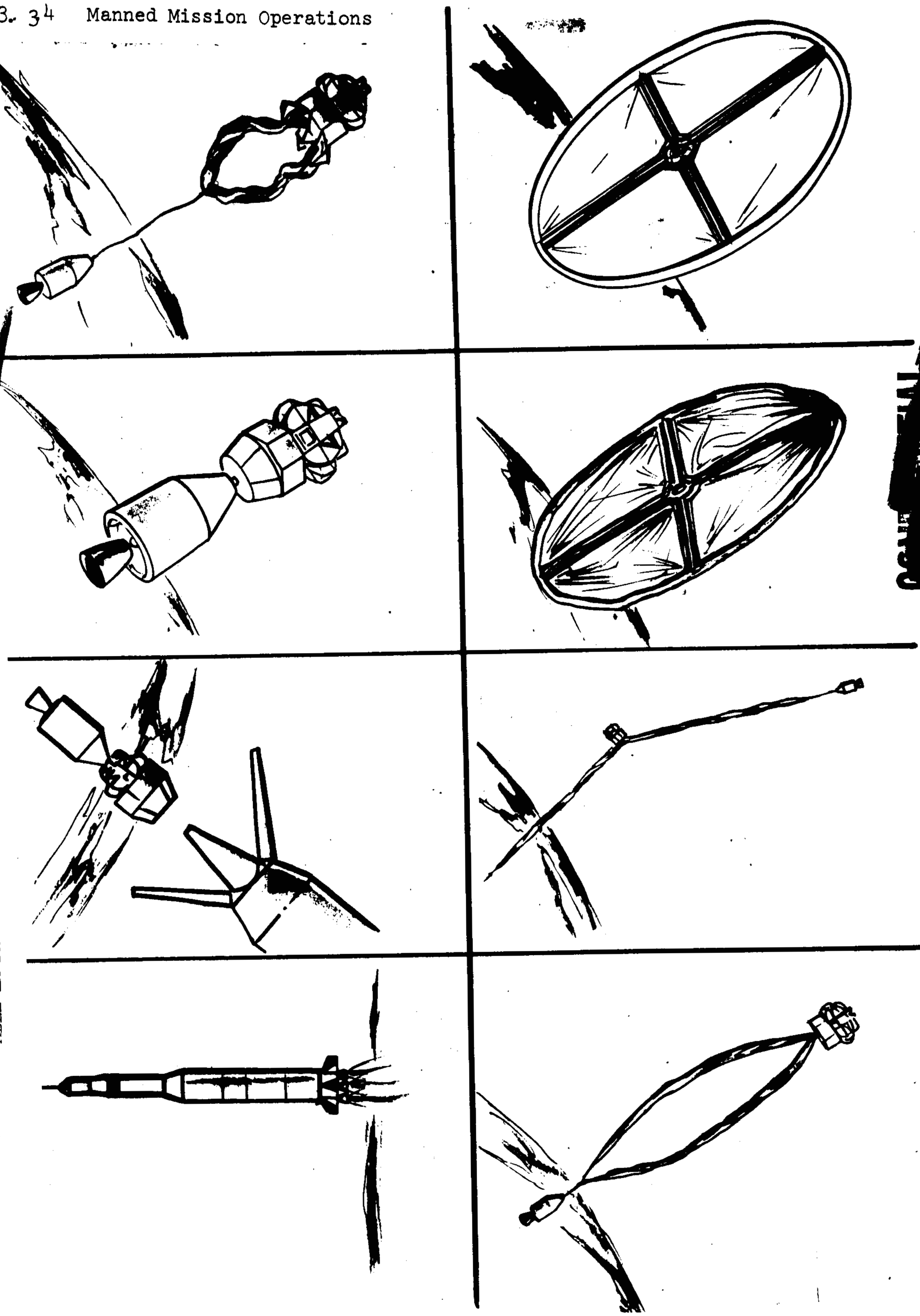
The results of the mission would demonstrate the Reflector deployment and rigidization systems in the space environment as well as the control capabilities of the Reflector subsystem. The atmospheric environment in low earth orbit is different from high altitude orbits in terms of disturbances. However, the basic

- deployment and rigidization method,
- control characteristics under orbital space conditions,
- applicability of man to support initial operations of large deployable structure

can be evaluated either directly or with well founded extrapolation. Furthermore, the presence of crewmen in monitoring the first deployment and operations will provide first hand substantiation of ABLE design practice or enable design modification of subsequent operational missions.

[redacted]

3.34 Manned Mission Operations



ABLE LM DEPLOYMENT CONCEPT

[redacted]

[redacted]

4.0 REFLECTOR/SUPPORT STRUCTURE

4.1 MATERIALS

Organic films were considered for the reflector material and for use in expendable support structure design. The principle properties of interest for this application are low and high temperature serviceability, tensile strength, elongation, creep, specific gravity, tear strength and radiation resistance. In addition, other factors such as fabrication, width and thickness availability and cost must be considered. Of the plastic films presently available, Mylar (polyester) and Kapton (polyimide) offer the best combination of these properties.

4.1.1 Reflector

Since it is recommended that the film be reflective on both sides it should be coated with vacuum deposited aluminum on both surfaces. The aluminum provides the added function of protecting the plastic film from the damaging effects of ultra-violet (U.V.) radiation. It is also recommended that the reflector be designed with no thermal control coatings. When this reflector is in sunlight, however, the temperature of the aluminized film can rise to about 400°F. This constraint necessitates the use of Kapton as the reflector material since 400°F is outside the useful temperature range of Mylar.

The minimum gauge of Kapton presently available is 0.5 mil but Dupont representatives indicate that experimental work on thinner gauges will be undertaken in 1967. It is anticipated that 0.35 mil will therefore be available for use on this program. Adhesives are presently being tested and developed by Schjeldah for bonding of the Kapton and there are indications that satisfactory adhesives will shortly be available.

Creep of the Kapton could cause dimensional changes beyond those compensated for in the reflector design. Although this magnitude of creep would be undesirable, a small amount of creep was found to be suprisingly beneficial, since it eliminates wrinkles in the reflector caused by fabrication and packaging. These wrinkles represent stress concentrations, and when the material creeps

[redacted]

4.1.1 cont'd

these areas are the first to be relieved. The applied stress should be the minimum value which eliminates these wrinkles in a short time (about 1 day at 400°F) and yet not cause excessive creep in 6 months. Visual observations of 0.5 mil aluminized Kapton specimens under load at 400°F indicate that stresses of 50-200 psi relieve a majority of the wrinkles in one day.

4.1.2 Support Structure

A typical support structure consists of a composite of 0.5 mil aluminum -0.35 mil plastic - 0.5 mil aluminum. Since (a) thermal control coatings on the support structure keep the maximum temperature within the useful range of Mylar (b) manufacturing techniques for fabricating an aluminum-Mylar laminate have been developed and successfully used on the Echo II program and (c) Mylar is availabe in wider widths than Kapton, Mylar is preferred for use as the plastic in the laminate.

The proposed coatings on the torus are those used on the Echo II balloon (i.e. exterior alodine, interior-carbon black) as processing equipment and procedures have already been developed for the Echo program. The thickness of the aluminum makes alodine processing practical for the support structure, but not for the reflector.

The recommended approach to minimize the structural deflections caused by the temperature gradient is to wrap the spokes with layers of "superinsulation".

[redacted]

[redacted]

4.2 STRUCTURAL ANALYSIS

Structural analyses of two general configurations were performed.

- o a circular reflector membrane supported by a torus with four spokes
- o a triangular reflector of equivalent area supported by three spokes intersecting at the center of area

The basic material used for the structural supports for the above configurations is aluminum-mylar laminate, which is stressed above the yield during erection to obtain rigidization of the material. The aluminum layers are each 0.0005 in. bonded to a layer of 0.00035 in. mylar. The structural supports are twenty foot cylindrical tubes made of the above material stiffended by 5 inch diameter rigidized tubes in the form of stringers and frames. The reflector membrane is made of aluminized Kapton with a thickness of .00035 inches.

The effects of gravity gradient loads, solar pressure, thermal gradients, membrane tension forces, membrane deflections, natural frequencies and creep were studied. The results of this analysis indicate that the deflection of the overall system is primarily due to thermal gradients in the spokes, in addition to manufacturing tolerances. The effect of solar pressure and gravity gradients on deflections is small. The analysis also indicates that reflector membrane tensile loads are strongly influenced by the manufacturing tolerances; large tolerances can induce high tensile stresses.

In addition to the structural cross sectional arrangement mentioned above this investigation also included a study of a stabilized structure utilizing internal pressure. If a pressure of 2 psi could be maintained in the reflector support structure, the allowable bending moment in the torus and spokes for the twenty foot diameter section can be increased by a factor of 1.5 over the unpressurized structure. However consideration must be given to the weight of gas necessary to carried to maintain pressure lost due to meteoroid penetration. A study of this effect was made for the one year life requirement. The results of the calculation showed that approximately 1.5×10^6 lbs of pressurization gas would be necessary in order to maintain the required pressure. In view of the excessive weight, this structural system was not considered feasible.

[redacted]

[redacted]

4.2 (continued)

A structural efficiency study was also conducted on a photolyzable film wire mesh concept for a rigidized structure. A comparison using published compression data was made for a photolyzed wire mesh cylinder and an aluminum foil cylinder. The result showed that the aluminum cylinder was capable of carrying a larger compression load than the wire mesh structure.

Finally, early consideration was also given to a support structure consisting of aluminum mylar laminate cylinders without internal pressure. However, this structure results in extremely large r/t values giving negligible values of bending stress allowables.

It is concluded from the analyses and studies carried out under this program that it appears structurally feasible to design and fabricate an expandable structure to carry out the assigned mission. The more promising structural arrangements are analyzed in the remainder of this report, i.e. the torus-spoke configuration and the tripod configuration. It must be noted, however, that problem areas still remain for further study. These include the following:

- bending and compression buckling test data
- more accurate analyses of the effects of manufacturing tolerances
- detail static and dynamic analyses of the effects of load and thermal disturbances
- materials creep data and creep deformations.

[redacted]

4.3 THERMAL ANALYSIS

4.3.1 Reflector

The choice of a reflector material is dependent on the temperature extremes that will be encountered. Reflector temperatures are dependent on the material thermal properties and on mission orbital parameters

The reflector thermal design concept consists of aluminum vapor deposited on both sides of a Kapton sheet (nominal surface $\alpha/\epsilon = 4.0$). This coating results in a nominal maximum reflector temperature of 395°F. This temperature is within the working range of Kapton (see Section 4.4).

4.3.2 Support Structure

For any given orbital position of the spacecraft, the heat flux distribution around a cross-section of the support structure will be non-uniform. This will lead to temperature differences within the structure and possible thermal distortion problems. The structural thermal design must be directed toward reduction of these temperature differences within the cylindrical cross-sections of the structure. A feasible thermal design concept follows.

The interior surfaces of the inflatable structure are coated with a "black" material having an emittance of 0.9. The spoke protion of the structure is wrapped with a super insulation blanket consisting of a number of layers of aluminized mylar to provide isolation from the external non-uniform environmental heat flux. For example, an insulation blanket consisting of a total of six layers of crinkled aluminized mylar used in conjunction with an internal surface emittance of 0.9 will reduce the temperature difference within the structure to a nominal value of 5°F with a range approximating ±2°F depending on factors such as handling, uniformity of material properties, etc.

[redacted]

5.0 ABLE LM

5.1 THERMAL ANALYSIS

5.1.1 Reflector Inflation

The reflector inflation system utilizes helium to deploy and rigidize the reflector support structure at a controlled rate. Initial deployment is accomplished using helium stored in two LM RCS helium tanks located on the periphery of the outer torus. After initial deployment, the support structure is inflated to strain rigidization pressure by helium stored in the LM Descent Propulsion System Cryogenic Tanks.

5.1.2 LM Equipment

All electronic equipment located aboard the ABLE LM vehicle will be temperature controlled in the same manner as in the present LM. The majority of equipments are mounted on a cold rail or cold plate through which glycol flows, absorbing the dissipated heat. Temperature control of externally located equipment such as the LM RCS jets and the LM S-band steerable antenna is achieved by existing thermostatically controlled heaters.

5.2 REACTION CONTROL

5.2.1 ABLE LM RCS

During deployment and checkout of the reflector, attitude and translational control will be required by the ABLE LM RCS. The basic sixteen thruster LM RCS will be retained. After completion of the deployment and checkout phase, the RCS would remain activated to support any stationkeeping requirements during the 6 month operational phase.

5.2.2 Peripheral Thruster System

Attitude control of the reflector would be achieved by means of small reaction control jets located at the periphery of the reflector. The choice for the peripheral thruster system, with total impulse requirements between 100,000 and 500,000 lb-sec for an 1100 ft diameter reflector, is between monopropellant and bipropellant.

Since the system weight difference is small, the simplicity, materials compatibility and reliability of the hydrazine monopropellant system indicate it to be the most likely selection. On the other had, if smaller reflectors were chosen, with total impulse requirements below 100,000 lb-sec, subliming rockets should be considered.

5.3 GUIDANCE AND CONTROL

5.3.1 System Requirements

An analysis of the Project ABLE mission identified the following major functions which the GNCS must provide.

- Attitude Hold Function
- Manual Control Function
- Alignment Function
- Pointing Function
- Orbit Keeping Function
- Remot Control Function - to redesignate the aim point of the reflector.

In order to provide the functions defined above, several alternate system concepts were considered. These ranged from the utilization of LM equipment, to the synthesis of a completely new system. The alternate systems considered were:

- Existing LM PGNCS (IMU, AOT, LGC, ATCA) with MSFN
- Modified LM PGNCS (IMU, LGC, ATCA, LM Optical Tracker System (LOTS)) with MSFN.
- Star Tracker Systems
- Gimballed Star Trackers, Sun Sensor, MSFN, Computer Systems
- Sun Sensor/Target Sensor Systems-Sun Sensor, ground beacon (at target), on board beacon sensor, Computer, MSFN

Based on the characteristics of the systems described above, the LM PGNCS configuration with the addition of the LM Optical Tracking System (LOTS) is generally recommended. The fundamental reasons for this decision are:

- The PGNCS - LOTS configuration can perform the ABLE mission
- All other systems require either design and development of new hardware, utilize unproven systems concepts, or would present seriour interface problems if incorporated into existing LM subsystems

The LM PGNCS is an integrated, aided, inertial Guidance, Navigation and Control System. The basic components of the system, which would be used for the ABLE

5.3.1 (Continued)

GNCS function, consist of a stable platform (IMU) to provide an inertial reference, a star tracker system (LOTS) to realign the stable platform, a digital computer, (LGC) which provides the guidance and navigation function and contains the digital autopilot (DAP), astronaut hand controllers for rotational and translational maneuvers (ACA and T/TCA), a data entry device (DSKY), and an interface with reaction jet thrusters through the jet driver preamplifiers in ATCA.

An analysis of the ABLE GN&C requirements indicates that with suitable modifications the LM PGNCS can perform the GN&C functions for the ABLE mission.

The LM PGNCS is interfaced with a Reaction Control System (RCS) which is used to provide translational motion in response to guidance system thrusting commands and rotational motion in response to control system commands. When the reflector is not deployed, or is in the process of deployment, the RCS can be used to provide control torques and translational motion.

When the reflector is deployed, however, the reflector probably cannot absorb the relatively high torques associated with the RCS. Based on a preliminary study of various control torquers it was decided that reaction control thrusters mounted on the periphery of the reflector are probably best for attitude control.

5.3.2 Subsystem Description

Hardware modifications required for the ABLE mission are:

- Incorporate the LM Optical Tracker System LOTS
- Incorporate a modified Program Coupler Assembly (PCA)
- Provide an interface and command signals to a low level peripheral thruster system.
- Provide an interface and command signals to an EPS Solar Array and to the S-band antenna (as required).

In addition to the hardware modifications, the LGC Programs will have to be modified to satisfy the ABLE Guidance and Control requirements.

5.4 ELECTRICAL POWER

5.4.1 EPS Configuration

For the selected ABLE LM EPS configuration the existing LM EPS was retained and supplemented, for the six month unmanned phase, with a solar cell array and rechargeable battery power supply. The existing LM EPS, which uses primary batteries as the energy source, supplies the ABLE LM power requirements until the solar arrays are deployed. From that time to the end of the manned phase, the solar arrays and the LM batteries are operated in parallel to meet the relatively high power requirements during manned ABLE LM operations. During the six month unmanned phase, power is entirely supplied by the solar array and secondary battery power supply.

The supplementary power supply includes two silicon solar cell arrays, one on each side of the reflector, two secondary NiCd batteries (one for redundancy), battery charger assemblies, and a voltage regulator to maintain the power source output voltage within allowable voltage limits.

For a completely unmanned mission the EPS configuration remains the same with the exception that the manned control and switching requirements are replaced by remote uplink command.

5.4.2 Alternate Configuration

A survey of other types of power supplies capable of performing the ABLE mission was also made. The only other power supply which appeared to have the development status, power delivery capability, weight and configuration which could be considered for this mission was the SNAP 10A system.

The utilization of a SNAP 10A power supply appears to be feasible for the ABLE mission. This generator has the nominal power and life required for this mission and has been successfully flown in space for a period of 45 days. The system would be mounted on the LM in a shutdown state prior to launch, and, after reflector deployment and vehicle alignment, the SNAP 10A would be deployed on a boom. In a manned mission, start-up would occur after the astronauts have left the vehicle.

5.5 ENVIRONMENTAL CONTROL

The only significant differences between the LM and ABLE LM ECS configurations are those associated with satisfying the requirement for active heat rejection during the six month operational phase of the ABLE mission. This functional capability has been achieved through the integration of radiator panels and associated controls, eliminating requirements for expendible evaporants during the six month period.

The following LM ECS sections were retained essentially unmodified:

- Atmosphere Revitalization Section
- Oxygen Supply and Pressurization Control Section
- Water Management Section

The modifications to the Heat Transport Section resulted in the following hardware changes:

- Addition of two 30 ft^2 radiator panels located 180° apart on the sides of the Descent Stage
- Addition of control valves (Apollo/LM hardware)
- Addition of a regenerative heat exchanger (LM hardware)
- Change coolant from 35% glycol to 62½% glycol (CSM coolant)

5.6 COMMUNICATIONS

5.6.1 Selected Configuration

The ABLE LM Communications would be composed almost entirely of LM developed hardware.

The following equipment was added to the basic LM Communications subsystem to obtain the ABLE LM configuration:

- Command Receiver - The Command Receiver will interface the DCA (LM 1 Modified) to turn on power and provide the uplink data chain. In addition, other on-off functions may be required of the Command Receiver.

[redacted]

5.6.1 cont'd

- Modified Digital Command Assembly - The DCA is modified to accept the Command receiver data chain output. The function of the DCA remains the same as on LM 1, and the output interfaces remain unmodified (PCA and LGC).

During manned portions of the mission the communications subsystem provides the voice links between the ABLE LM occupant(s) and EVA, the CM, and Earth. Bio-med data of one occupant and/or an EVA can be transmitted to earth. Sapcecraft data (PCM at 51.2 Kb/s) is also sent to earth. PRN ranging capability exists, but is not used during the manned phase as the CSM would provide this function. In general, the subsystem operates as on LM.

During the unmanned portion of the mission, the Communications Subsystem is the ground control link. Updating information and commands are received, decoded and routed to the various vehicle equipments. In addition, PRN ranging data is obtained to aid in orbital parameter, and spacecraft data is sent to ground.

The updata link is designed to operate independently from the LM type equipment. For example, an uplink message can be received, decoded, and executed without the downlink being turned on. For verification of commands, the downlink must first be commanded on.

5.6.2 Alternate Configuration

The selected configuration utilized the LM S-band uplink/downlink capability. Another configuration considered was the UHF uplink, VHF/S-band downlink which fundamentally retains the LM 1 UHF command uplink along with the VHF downlink (for low altitude mission phases if required). If necessary, backup operational modes would utilize the S-band downlink. If desired the C-band transponder system provides a ranging/tracking aid independent of the S-band system.

[redacted]

5.7 INSTRUMENTATION

The recommended Operational Instrumentation Section for the Able LM is comprised of existing LM assemblies having a certain built-in flexibility which allows for some minor configuration changes. A measurements list prepared for the standard LM vehicle was reviewed and analyzed to assure that the changes to support the requirements of ABLE LM would still be adequately covered using the existing LM system.

5.8 CONTROLS & DISPLAYS

A detailed analysis of Control and Display requirements for the ABLE LM is dependent upon better definition of specific crew tasks and reflector-related sensors. However, the extent of potential modifications to the LM Controls and Displays is considered to be minimal.

5.9 CREW PROVISIONS

The interior of the ABLE LM Ascent Stage will probably be similar to the Apollo LM, with minor modifications associated with the Control & Display consoles.

Equipment required for EVA can be retained as in the existing LM configuration. The existing LM food and waste management system can be retained for possible manned ABLE LM operations while separated from the CSM.